AD-A014 312

COMPARISON OF TURBOJET, TURBOROCKET AND RAMJET AS A PROPULSION SYSTEM FOR LONG RANGE AIRPLANES AT MACH NUMBERS BETWEEN 2 AND 4

E. Riester

Foreign Technology Division
Wright-Patterson Air Force Base, Ohio

31 July 1975

FTD-ID(RS)I-1653-75

FOREIGN TECHNOLOGY DIVISION

COMPARISON OF TURBOJET, TURBOROCKET AND RAMJET AS A PROPULSION SYSTEM FOR LONG RANGE AIRPLANES AT MACH NUMBERS BETWEEN 2 and 4

by

E. Riester

Approved for public release;
distribution unlimited.

UNCLASSIFIED
Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) Foreign Technology Division Air Force Systems Command U. S. Air Force	2a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED 2b. GROUP

3. REPORT TITLE

COMPARISON OF TURBOJET, TURBOROCKET AND RAMJET AS A PROPULSION SYSTEM FOR LONG RANGE AIRPLANES AT MACH NUMBERS BETWEEN 2 AND 4

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

Translation

5. AUTHOR(S) (First name, middle initial, last name)

E. Riester

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
1972	31	19

8a. CONTRACT OR GRANT NO.	9a. ORIGINATOR'S REPORT NUMBER(S) FTD-ID(RS)I-1653-75
b. PROJECT NO.	
c.	9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report)
d.	

10. DISTRIBUTION STATEMENT

Approved for public release; Distribution unlimited.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY
	Foreign Technology Division Wright-Patterson AFB, Ohio

13. ABSTRACT

21;01

PRICES SUBJECT T...

i-a

DD FORM 1 NOV 65 1473

UNCLASSIFIED
Security Classification

EDITED TRANSLATION

FTD-ID(RS)I-1653-75 31 July 1975
FTD-75-C-002117

COMPARISON OF TURBOJET, TURBOROCKET AND RAMJET AS A PROPULSION SYSTEM FOR LONG RANGE AIRPLANES AT MACH NUMBERS BETWEEN 2 AND 4

By: E. Riester

English pages: 31

Source: Deutsche Luft- Und Raumfahrt, Forschungsbericht 72-38, 1972, pp. 1-36.

Country of origin: West Germany
Translated by SCITRAN
F33657-72-D-0853
Requester: ETIL
Approved for public release; Distribution unlimited.

THIS TRANSLATION IS A RENDITION OF THE ORIGINAL FOREIGN TEXT WITHOUT ANY ANALYTICAL OR EDITORIAL COMMENT. STATEMENTS OR THEORIES ADVOCATED OR IMPLIED ARE THOSE OF THE SOURCE AND DO NOT NECESSARILY REFLECT THE POSITION OR OPINION OF THE FOREIGN TECHNOLOGY DIVISION.	PREPARED BY: TRANSLATION DIVISION FOREIGN TECHNOLOGY DIVISION WP-AFB, OHIO.

SUMMARY

In the Mach number range between 2 and 4, a transition to another propulsion system is expected for long range airplanes. Limited to the cruise range,the turbojet, the turborocket and the ramjet are investigated. Considering also the additional drag, engines are compared, whose air flow rates have the same cross section area for the undisturbed flow in front of the engine. It is shown that even with modern component technology, the turbojet is the optimum propulsion system only up to the Mach number 3.5. Above this Mach number, the ramjet becomes more effective. The turborocket is interesting at high Mach numbers because of its high thrust density, although its specific impulse is somewhat less.

TABLE OF CONTENTS

NOTATION LIST

c_s	Thrust coefficient	
c_w	Drag coefficient	
$C_{10}H_{20}$	Chemical formula selected to represent kerosene	
h	Enthalpy	J/kg
h_1	Enthalpy ahead of the turbine	J/kg
h_2	Enthalpy after isentropic expansion	J/kg
M	Mach number	
$O_{2\ min}$	Oxygen requirement for the stoichiometric reaction	
p	Pressure	bar
ST	Ramjet engine	
T	Temperature	K
TL	Turbojet without afterburner	
TLN	Turbojet with afterburner	
TR	Turborocket	
w	Velocity	m/sec
η_T	Turbine efficiency	

Subscripts

Boden	In the vicinity of the Earth's surface
e	After decompression in the turbine of the turborocket
F	Surroundings, flight state
G	Gas generator
is	After isentropic decompression
crit	Critical state of tube flow
o	State at rest (also in front of other subscripts)
1-7	Stations in engine (see Figure 5)
8	Behind engine after completed pressure equilization

COMPARISON OF TURBOJET, TURBOROCKET AND RAMJET AS A PROPULSION SYSTEM FOR LONG RANGE AIRPLANES AT MACH NUMBERS BETWEEN 2 AND 4*

E. Riester

1. INTRODUCTION

The static temperature and pressure of the flow increases considerably because of stagnation in the higher supersonic range. In order to obtain an idea of the order of magnitude, Figure 1 shows the ratio of the pressure and temperature and the surrounding values for isentropic stagnation. It can be seen that at a Mach number of 4, the static temperature T_o has increased to about four times the surrounding temperature T_F and that the static pressure p_o has increased to 150 times the surrounding pressure p_F. For flight in the vicinity of the ground, the engine would have to be surrounded with a type of pressure chamber for 150 bar difference pressure, which is out of the question because of the high container mass. Therefore, one must attempt to reduce the pressure.

The usual way is to only allow high Mach numbers at high altitudes. Figure 2 shows a variation of the pressure and temperature as a function of flight altitude according to [1]. One can see the strong decrease of the pressure with altitude. The temperature first also decreases and reaches a minimum at an altitude between 11 and 20 km (about 217 K) and then slightly increases. As we will show, the altitude range between 11 and 20 km and somewhat above is of particular

* German Aerodynamics and Spaceflight Research Report 72-38.
** German Research and Test Facility for Aerodynamics and Spaceflight. Institute for Air Breathing Engines. Braunschweig.

interest. The following developments will be restricted to this range. The surrounding temperature can be looked upon as constant and as 217° K. In this way in the following, we can avoid a parameter variation of the surrounding temperature.

An additional pressure reduction in the engine, which is undesirable, is the result of the engine inlet. In spite of many attempts [2, 3, 4, 5 and many new publications], it has not become possible to completely win back the velocity energy in the form of pressure energy. Figure 3 shows the pressure gain for various optimum shock configurations for external and internal compression. Even for the isentropic diffusor, the pressure recovery is smaller than 1, because the transition to the subsonic speed by a shock must occur for stability reasons and because the flowing condition implies that the shock cannot be made arbitrarily weak. The solid curve corresponds to the quite favorable US standard values according to the formula

$$p_{01}/p_0 = 1 - 0.1\,(M_F - 1)^{1.5}$$

minus 10% for losses in the subsonic diffusor, losses because of boundary layers and safety reserves. This curve will be used as a basis in the following calculation. It can be seen that the static pressure is reduced at Mach number 4 by more than a factor of 2 because of the inlet.

Figure 4 shows the pressure in the engine which occurs behind the inlet. Curves of constant pressure (static pressure) behind the inlet are shown in the flight altitude — flight Mach number diagram. It can be seen that in the altitude range between 11 and 20 km, established earlier, the pressure behind the diffusor has the order of magnitude of the surrounding pressure on the ground. Therefore, there is no fundamental difficulty in designing the engine casing for this pressure and the pressure increase with Mach number does not have to be especially considered in the following discussion.

The lower curve in Figure 4 shows the variation of the static temperature as a function of Mach number. The temperature can be only slightly reduced by flight altitude, and cannot be reduced

at all by modifying the inlet. It can be seen that at Mach number 4, one is faced with a static temperature of over 900 k. This temperature will be the basis of a discussion in the following.

2. BASIS OF COMPARISON

Comparisons of various propulsion types for supersonic aircraft were already made earlier [6, 7, 8, 9. 10]. However, either they are not at the present level of technology or they cover other velocity ranges, or they are restricted to concrete mission requirements. In the following comparison we will consider the cruise flight phase of a long-range aircraft in the Mach number range between 2 and 4. The advantage of this restriction is that it becomes possible to present the results in a clear way and that they will be independent of the mission. One disadvantage is that as the range decreases and for the extended flow flight phases during the mission, one departs somewhat from reality in calculations.

The propulsion unit is assumed to be designed for the Mach number range between 2 and 4. The curves shown are therefore a series of design states.

It is assumed that the inlet is the same for all propulsion types for each Mach number. Also the same maximum temperatures are assumed for all of the propulsion types which are allowable from the point of manufacturing according to present-day methods.

The engine size is often defined by the air throughput through the engine. The air throughput is one possible means of comparison for turbojets, especially in the subsonic range. However, it depends greatly on Mach number in the case of the ramjet engine and is zero on the test stand. Therefore, we will use the cross section occupied by the air throughput in the undisturbed incident air flow as a basis of comparison for the engine size. This phase of comparison is conventional for ramjet engines [11] and also has the advantage that with a parameter defined in this way at the considered Mach number, in the first approximation, all the engines will have the same aerodynamic drag with respect to the external flow. If we exclude

the diffusor lips which would be inappropriate starting with Mach number 3 anyway, then the reference area also at the same time becomes the main bulkhead area, except for the sheath wall thickness, and it becomes especially simple to take into account the aerodynamic drag.

One disadvantage is that below the Mach number of 3, we have a somewhat increased fuel consumption and a somewhat reduced thrust which must be accepted. This is especially true for the ramjet engine, which, however, is uninteresting in this Mach number range as an aircraft propulsion unit, as we will show. Another disadvantage of the selected basis of comparison is that the air throughput for the turbojet differs at each Mach number and therefore the compressor and turbine designs will be different. However, this is not detrimental because the engines are assumed to be designed for any Mach number, as already mentioned, and no behavior over a range is assumed.

3. SPECIAL FEATURES OF THE ENGINE TYPES

Figure 5 shows the three considered engine types in a schematic diagram. As already mentioned, the F-1 inlet is the same for all engine types. The afterburner chamber and nozzle 4-7 are the same in the diagram, but the thermogasdynamic data and the nozzle throat cross sections are different.

In the case of the ramjet engine, the afterburner connects to the inlet. The path 1-4 has a certain length only in the diagram.

In the case of the turbojet (tcp) the turbopart is switched between 1-4, consisting of the compressor 1-2, combustion chamber 2-3 and turbine 3-4. The turbopart has a constant main bulkhead area only in the diagram model. In reality, a cross section results as usual from the air throughput and the requirements of the blading. In the following we will have to discuss this further. Instead the turbopart will be calculated in the form of a thermogasdynamic cycle process calculation without velocity consideration, but we will consider the achievable efficiencies.

In the case of the turbojet (center), the incoming suction air is also compressed in a compressor 1-2. The drive of the compressor is done by a turbine which has a much smaller diameter, which is driven by the exhaust gases of a gas generator G. The turbine exhaust gases mix with the air throughput along 2-4. Because of the diameter difference, there must be a gear between the turbine and the compressor.

3.1. Turbojet

An efficiency of 88% is assumed for the compressor and it is 86% for the turbine. Losses in the combustion chamber are ignored. The permissible turbine inlet temperature must be assumed as high as possible, so that sufficient heat can be supplied in the combustion chamber at the high compressor inlet temperature (see Figure 4). Modern studies [12] and extrapolations for the middle of the 70's [13] consider turbine inlet temperatures of 1600 K to be permissible.

The possibility of supplying heat to the combustion chamber depends on the compressor pressure ratio. Figure 6 shows the possible enthalpy increase, referred to the compressor exit enthalpy for various compressor pressure ratios.

The requirement of having a temperature increase of 600 K in the combustion chamber of Rolls Royce [13] is represented in Figure 6 approximately by a horizontal line at 0.6. It can be seen that high flight Mach numbers require a low compressor pressure ratio. If the compressor pressure ratio is too low, then the characteristics of the turbojet approach those of a ramjet engine, and the blading is soon found to be a disadvantage. If we consider the compressor pressure ratio 5 to be used in the following discussion, we approach the value of the ATAR 9 C, the engine of the Mirage V, conceived as a pure interceptor aircraft. The pressure ratio 5 is disadvantageous during the takeoff, landing and slow flight phases. In the case of other supersonic engines (for example J 79 in the Starfighter, Phantom), the compressor pressure ratio is therefore higher and represents a compromise between the requirements of supersonic flight and the extended subsonic missions, a compromise caused by mission considerations.

The turbine of the turbojet delivers the drive power for the compressor. Figure 7 gives an idea of the power requirement for various compressor pressure ratios. Figure 8 shows the fraction of heat supplied to the combustion chamber with respect to the energy throughput (supplied heat plus compressor work) through the combustion chamber. The distance between the curves up to the upper edge of the figure is the compressor work fraction of the energy throughput. It can be seen that the compressor work fraction increases greatly with increasing flight Mach number and increasing compressor pressure ratio and soon reaches a value of 1. This is also the reason for selecting a low compressor pressure ratio.

3.2. Turborocket

In the case of the turborocket, the air throughput does flow through the compressor but not through the turbine. Therefore, the air temperature limitation caused by the turbine inlet temperature does not exist. Therefore, an expensive price must be paid for the compressor drive: the turbine is driven by the exhaust gases of a gas generator, whose propellant consists of the fuel plus the oxidizer. The specific fuel consumption is therefore relatively high and the specific impulse is low. The unit cost of the fuel is therefore higher than for the turbojet.

In order to save drive power, the compressor pressure ratio should be low, especially lower than for the turbojet (see also Figure 7). In [9], a two-stage compressor with a compression ratio of 2.5 is suggested. In [14], a two - stage compressor with a static pressure ratio of 2.3 (test stand on the ground) is suggested. In [12] a single stage compressor with a pressure ratio of 1.6 is suggested. In the following investigations, we will select a compressor pressure ratio of 2.0 which is considered to be an average value of the suggested values. For a smaller compressor pressure ratio, the characteristic parameters of the turborocket approach those of the ramjet engine, and for larger compressor pressure ratios, they deviate from them.

The propellant selection is very important for turborockets. Monergols such as hydrazine or hydrogensuperoxide have insufficient energy; they result in turbine inlet temperatures which are too low and therefore too high propellant consumptions. A combination between hydrogensuperoxide and fuel (kerosine) was investigated in detail in [12]. In order to avoid excessively high turbine inlet temperatures, the gas generator is operated there with a lean mixture. In this way a number of problems considered in the following are eliminated. Disadvantages include a poor specific impulse of the turbojet and an increased unit cost for the propellant compared with the version now being described.

For large ranges, the gas generator should be operated with a high energy, storable and cheap fuel (kerosine) and an oxidizer which has the lowest mass possible, referred to the oxygen component. The most favorable oxidizer is liquid ozygen. In order to reduce the turbine inlet temperature, the fuel-oxygen ratio should not be stoechiometric. Therefore, an excess in fuel is recommended, because it can be used in the afterburner and does not increase the specific propellant consumption. In [9] and [14] a fuel-oxidizer ratio of about 5 is suggested, i.e., 20% of the fuel could be stoechiometrically oxidized and 80% is used for cooling.

In practice, the reaction in the gas generator is more complicated. The combustion products and their thermodynamic state variables are obtained using the chemical reaction equations and the equilibrium constants for all of the reasonable end products while considering dissociation. This determination requires a great deal of time. There is a computer program at the computer center, Darmstadt with which the calculation can be performed [19]. It is possible to reduce the computational effort by using diagram representations [15, 16, 17] which does not require the use of such a computer program.

The result of the calculation is shown in Figure 9. The reaction occurs for the abscissa value one. To the left of this, one can see the mass fractions before the reaction for a 5-fold kerosene excess (oxygen requirement O_{2min} = 3.108 kg/kg kerosene), and on the right one can see the composition of the combustion products. The

kerosene has been completely split up. The main component is carbon monozide (CO) and in addition there is a substantial fraction (23%) of carbon in the form of soot. The mentioned computer program also provides the data after isentropic relaxation in a nozzle. These data are shown in Figure 9 for chemical equilibrium up to 1 thousandths of the gas generator pressure. The carbon fraction also increases during the relaxation.

The carbon fraction is undesirable from 2 points of view: first of all there is a danger that the turbine flow will be disturbed because of soot deposits. Also it is difficult to oxidize the carbon in the afterburner: the afterburner efficiency drops and the carbon reaches the atmosphere as a pollutant. Therefore, one must attempt to oxidize the carbon as much as possible in the afterburner chamber. If this is not possible, then the carbon fraction in the exhaust gas can be reduced by using another hydrocarbon instead of kerosine which contains more hydrogen compared with carbon. Another possibility is the use of a different oxidizer: it can chemically bind carbon and can absorb heat because of its higher mass. In this way the fuel excess can be reduced. Finally, water can be ejected into the gas generator for cooling, which also brings about a reduction in the combustion excess.

In the following we will not further discuss the problem of carbon expulsion. For the thermodynamic calculation, it is assumed that the carbon in the turbine does not have any detrimental effects and that it does not have a contribution to the heat supply in the afterburner.

Figure 9 shows three further abscissa scales. The relaxation temperature is coupled with isentropic relaxation and the enthalpy difference is converted in the nozzle into velocity in the isentropic case. Now the relaxation is not being performed in a nozzle but in a turbine, and it is not isentropic but there is an associated efficiency η_T. Under the assumption that the reaction processes are pressure-independent, which is satisfied within the accuracy of representation, according to Figure 10, it is possible to find a pressure ratio p_o/p_e for each turbine efficiency which has the same

enthalpy h and therefore the same temperature and gas composition as the isentropic relaxation pressure ratio p_o/p_{is}:

$$\eta_T = \frac{h_1 - h_2}{h_1 - h_e} .$$

For 86% turbine efficiency, the required turbine pressure ratio in Figure 9 is shown as the lower abscissa scale.

We select a pressure ratio of 16 for the turbine of the turbo-rocket. This corresponds approximately to the isentropic pressure ratio of 10 and a work output of 1.9 MJ per kg gas generator exhaust gas. One obtains the required size of the turbine and of the gas generator from the performance uptake of the compressor. The turbine exhaust gases in general are somewhat warmer than the air behind the compressor: the air throughput is slightly heated during the mixing. The momentum of the exhaust gases in the turbine exhaust is ignored.

3.3. Ramjet engine and afterburner

As already mentioned, the ramjet engine can be looked upon as a combination of inlet and afterburner chamber. Therefore, we will immediately discuss the afterburner. The combustion efficiency in the afterburner is assumed to be 85%. The fuel amount must therefore be accordingly increased in order to equalize the losses and to reach the desired temperature. In the case of the turborocket, there are additional losses caused by the uncombusted carbons. The thermogas-dynamic pressure loss [18] is considered. For this purpose, we must determine the flow velocity at the point 4 (Figure 5). An additional pressure loss caused by combustion chamber inserts, such as flame supporters, etc., is ignored.

The heat supply occurs always up to the same maximum ambient temperature, unless thermal blocking occurs before this, such as in the case of the ramjet engine below Mach number 2.8. The maximum temperature is specified at 2,000 K, which is permissible for the engine J 79. The ramjet engines often have higher maximum temperatures up to 2400 K, especially in units for short time missions. For the

turborocket, these values lie between 2100 K [12] and 2300 K [9]. In the following comparison, we would only allow 2,o00 K for these two engine types as well.

The nozzle throat must always be designed according to the continuity equation for the combustion chamber pressure. The nozzle end is extended to the outside diameter of the engine.

4. RESULTS

We carry out a thermogasdynamic comparison calculation on the computer using the assumptions made in the previous sections. We will use specific heats which are constant within sections, unless something different was specified in the previous discussion. In the following we will discuss the results.

Figures 11-13 show the variations of the thermogasdynamic state variables in the engine. The characteristic points are connected by straight lines. The temperature variations of Figure 11 essentially reflect the assumptions made: all the curves begin at the static surrounding temperature of 217 K. There is a temperature increase at the inlet 1-2 which depends on Mach number, but it is the same for all of the engine types. At 1 the curves divide. The highest temperature in front of the afterburner occurs for the turbojet (solid curve) at the combustion chamber outlet point (1600 K). The curves for the turborocket (TR, dashed curves) and for the ramjet engine (ST, dash and dot curves) are below it. In the afterburner chamber (after 4), the curve for the turbojet divides into a lower branch (TL, without afterburner) and an upper branch (TLN, with afterburner) At the afterburner outlet point, the temperature in most cases reaches a maximum position permissible temperature of 2000 K ambient temperature. The static temperature is somewhat lower and decreases substantially in the following expansion nozzle 5-7.

In addition to the turbojet without the afterburner, in the case of the ramjet engine for M_F = 2.5 (upper part of the figure), the afterburner ambient temperature (point 5) is smaller than 2000 K: because of the thermal blocking,the heat supply has to be reduced.

Figure 12 shows the pressure variations in the engines. One remarkable feature is the relatively high pressure load of the turbojet in the turbopart. Behind the engine outlet point (7) there is a pressure equilization with the surroundings. It can be seen that the nozzle is always operated at a counterpressure which is too low, and therefore, the extension of the nozzle to the engine outside diameter is advantageous.

Figure 13 shows the velocity variation. The velocity according to calculation is zero in the turbopart, because the turbopart is only described thermodynamically. In reality, there must be an axial velocity corresponding to the air throughput and the blading. However, it is relatively low and can hardly be represented within the selected ordinate scale. In the case of the ramjet engine (ST), the velocity in the nozzle (5-7) is constant for $M_F = 2.5$. Here, we must dispense with the inclusion of the nozzle throat because of the thermal blocking which would occur anyway.

Behind the engine outlet (7), it does not seem appropriate to give the velocity in one dimension. However, a replacement velocity w_8 can be given which considers the thrust part of the pressure term p_7-p_8 for the same flow tube cross section. Since the pressure term is positive, w_8 becomes greater than w_7. The velocity w_8, however, does not occur in reality. For this definition w_8-w_F is a measure for the engine thrust. However, we will not discuss the thrust here.

The last three figures give information on the performance capacity of the engine types. Figure 14 shows the specific impulse. It corresponds to the reciprocal of the specific propellant consumption. It can be seen that the turbojet is very favorable. It is only in the highest Mach number range (above 3.8) that it is necessary to turn on the afterburner and it is only in this region that the ramjet engine is somewhat better. The turborocket is clearly below these. Above a Mach number of about 3.3, its curve forks, the lower branch applies for the afterburner chamber temperature limitation of 2000 K. For the upper branch we consider the fact that in the gas generator more kerosine is injected for cooling than required for combustion in the afterburner. Since an air excess is still present in the afterburner,

the temperature there increases above 2000 K. At a Mach number of 4, it increases to about 2400 K. For comparison purposes, we will consider the lower branch of the curve.

Many authors restrict their engine comparisons to the consideration of the specific impulse. However, in order to describe the performance capacity of an engine, a secondary quantity is of interest: the thrust. Figure 15 shows the thrust referred to the main bulkhead area and the stagnation pressure. We have plotted the thrust coefficient c_s. In contrast to the previous figure, the turbojet is unfavorable here. It can be considered in the discussion only because of the afterburner. Above the Mach number of 3.5, the ramjet engine is more favorable. Above M = 2.5, the turborocket is the best, even if we only consider the lower branch of the curve (2000 K afterburner temperature). For Mach numbers below 2.8, the ramjet engine is unfavorable because of the thermal blocking for the selected configuration.

In addition to the thrust production and propellant consumption, an engine installation also results in an additional aerodynamic drag, which must be considered especially at high Mach numbers. The aerodynamic drag consists of wave drag, wall friction drag, base drag and interference drag. It depends greatly on the configuration. As an average value we will assume that an additional aerodynamic drag coefficient of $c_w = 0.2$ exists, referred to the main bulkhead area of the engine. In order to obtain the net thrust coefficient, it is necessary to subtract c_w from c_s. We then obtain a figure as shown in Figure 15, but with the ordinate scale displaced by 0.2.

The net thrust coefficient can be used to determine the net specific impulse shown in Figure 16. For comparison purposes, we also show the specific impulse of Figure 14 by means of a thin line. Of course the net impulse curves are all lower. It can be seen that the decrease for the turbojet is especially drastic. The ramjet engine maintains its favorable position starting at about M = 3.5. The decrease is relatively small for the turborocket. It can still be considered because of its high thrust density (see Figure 15) even

though its net specific impulse does not reach those of the ramjet engine or of the turbojet with afterburner at the high Mach numbers.

5. SUMMARY

We compared the turbojet, turborocket and ramjet engine as propulsion units for a long range aircraft during the cruise flight phase in the Mach number range between 2 and 4. In addition to the specific impulse, we considered the comparison of the thrust density and aerodynamic drag. It was found that the turbojet is the most favorable means of propulsion up to a Mach number of 3.5. Above this Mach number, the ramjet engine is the most favorable. The turborocket can be considered in the discussion because of its high thrust density, in spite of its high specific impulse. In the case of the turborocket, we should investigate questions associated with the gas generator, the turbine operation, the exhaust gas composition and the exhaust gas mixing.

6. REFERENCES

[1] US Standard Atmosphere. US Government Printing Office, Washington 25, D.C., (1962).

[2] FABRI, J. (Herausgeber) Air intake problems in supersonic propulsion. AGARDograph Nr. 27. Pergamon Press, London/New York/Paris/Los Angeles (1958).

[3] SÖFFKER, E. Beitrag zur Frage des Lufteinlasses bei Staustrahltriebwerken für Überschallgeschwindigkeit. Jahrbuch 1957 der WGL, S. 107-117.

[4] KRAUSE, E. Entwurfsprobleme supersonischer Einlaufdiffusoren. DVL-Bericht 198, Juni 1962.

[5] TRIEBNIGG, H.
SCHMIDT, G.
Zur rechnerischen Erfassung der Einlaufverluste bei Gasturbinen und Staustrahltriebwerken, insbesondere im Uberschallflug.
Luftfahrttechnik 6 (1960), S. 369-374.

[6] HERTEL, H.
Überschall- und Hyperschall-Verkehrsflugzeuge, Teil II: Das Triebwerk.
Flugwelt 1960, Nr. 6.

[7] LUTZ, O.
Möglichkeiten der Anwendung luftatmender Antriebe bei Raumfahrtprojekten.
2. Lehrgang für Raumfahrttechnik der DGF/WGLR (1963).

[8] MARGUET, R.
Choice of propulsion mode.
O.N.E.R.A.-Report N 71-29569.

[9] VOSS, N.
SINGELMANN, D.
Untersuchungen über die Möglichkeiten von zukünftigen Hochgeschwindigkeits-Höhenfluggeräten, Teil III.
EWR-Bericht 174/III (1968).

[10] MBB
Hochgeschwindigkeitstechnologien im Flugzeugbau. Zelle und Antrieb für Mach 3 bis 5, Teil I: Entwurfsuntersuchungen und Antriebsstudie.
ZTL-Bericht UFE 607/1-70, BMVtg Auftr. Nr. T/0135/04100/02041.

[11] FERRI, A.
Possible Directions of Future Research in Airbreathing Engines.
Fourth AGARD Colloquium, Combustion and Propulsion, Milan 1960, Pergamon Press, Oxford, London, New York, Paris (1961), S. 3-15.

[12] WEILER
u. andere
Untersuchungen zu Kombinationstriebwerken für sehr große Geschwindigkeiten und Höhen.
MTU-Bericht 71/062.

[13] HOOKER, S.G.
The engine scene.
Astron. Journal 74 (1970), Nr. 709, S. 1-11.

[14] LOMBARD, A.A. KEENAN, J.G. — Ein Turboraketentriebwerk für luftatmende Hochgeschwindigkeitsflugzeuge. Interavia 1965, Nr. 1.

[15] REICHERT, H. — Enthalpy charts for dissociating gas mixtures in the temperature range 600 to 4000 deg K. Ministry of Supply TN R.P.D. 35, August 1950.

[16] REICHERT, H. — High temperature combustion. Aircraft Engineering 25, No. 293 (1953), S. 198-212.

[17] LUTZ, O. — Technische Reaktionsthermodynamik. Z. für Flugwissenschaften 3 (1955), Heft 6, S. 151-159.

[18] LUTZ, O. — Diagrammdarstellungen der Vorgänge in Brennkammern und Staustrahltriebwerken. Jahrbuch 1955 der WGL, S. 252-265.

[19] ZELEZNIK, F.I. GORDON, S. — A General IBM 704 or 7090 Computer Program for Computation of Chemical Equilibrium Compositions, Rocket Performance and Detonations. NASA TN D-1454 (1962)

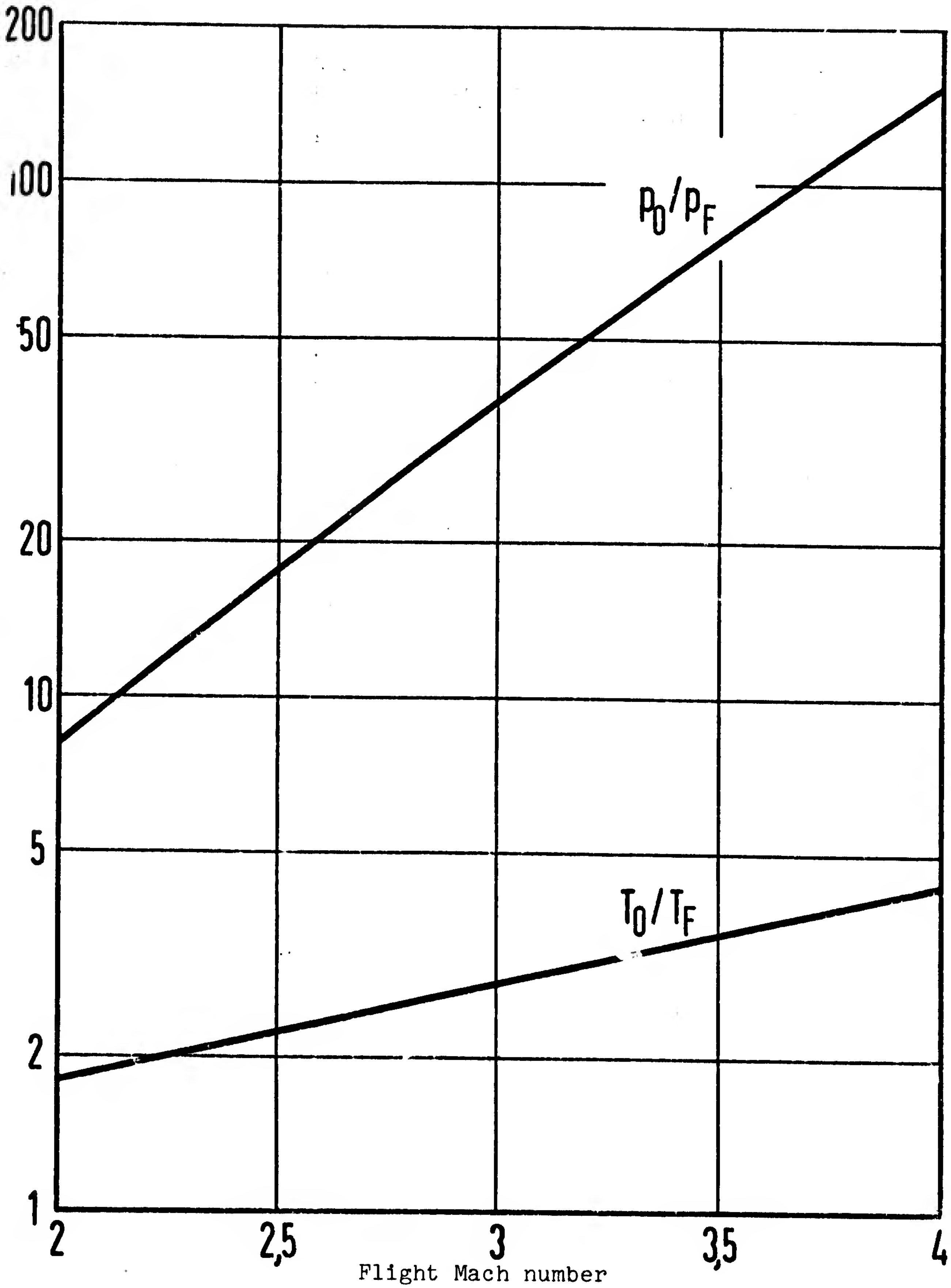

Figure 1. Increase of pressure and temperature with Mach number.

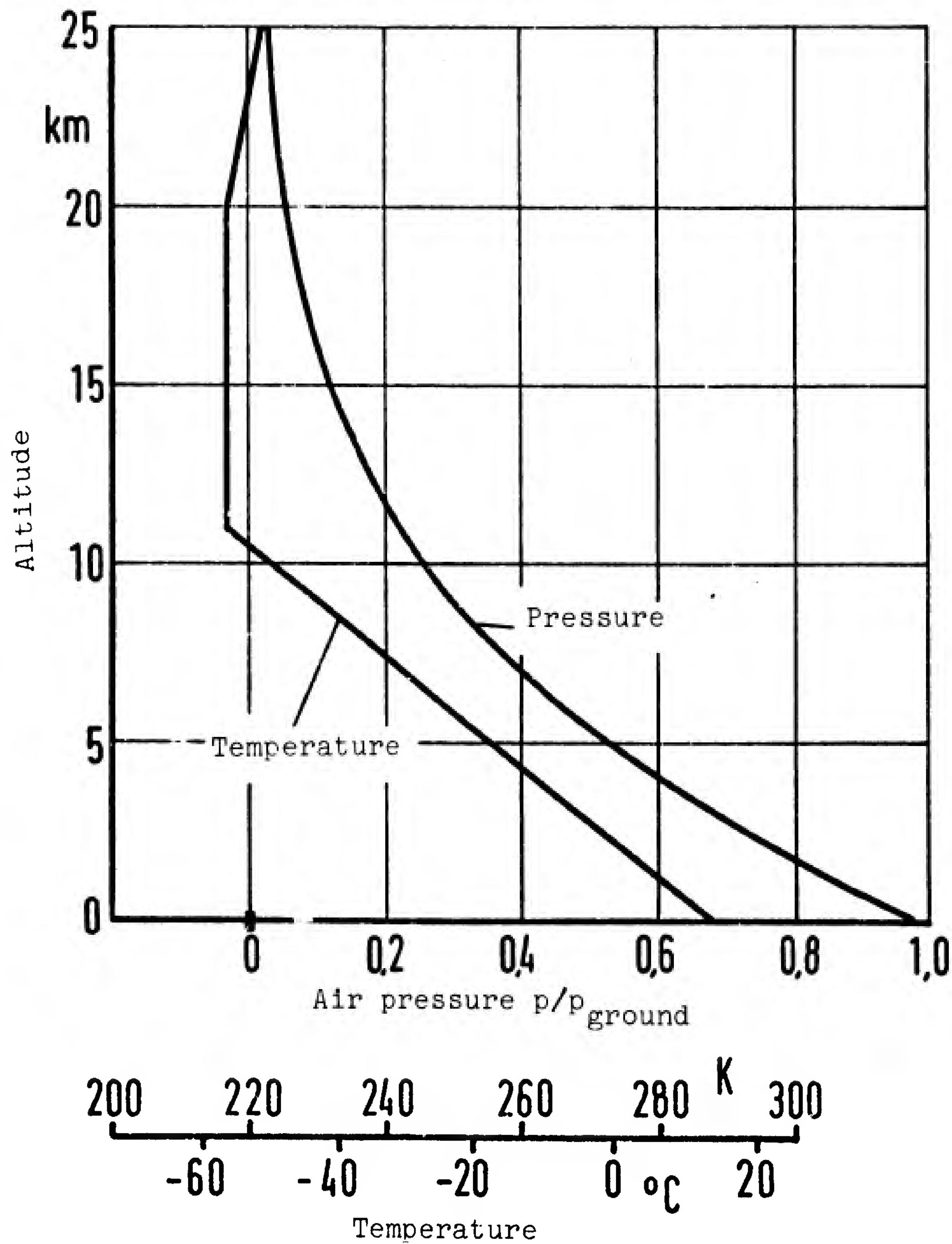

Figure 2. Dependence of pressure and temperature on altitude.

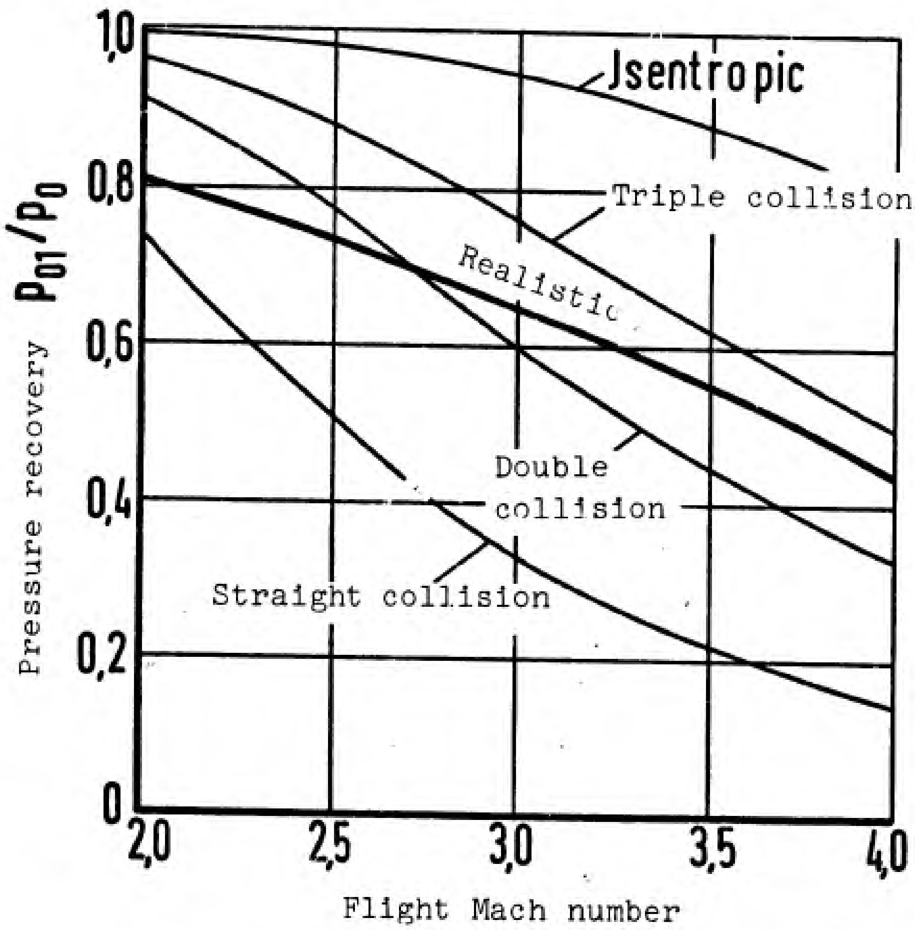

Figure 3. Pressure recovery in diffusor. Optimum internal and external compression.

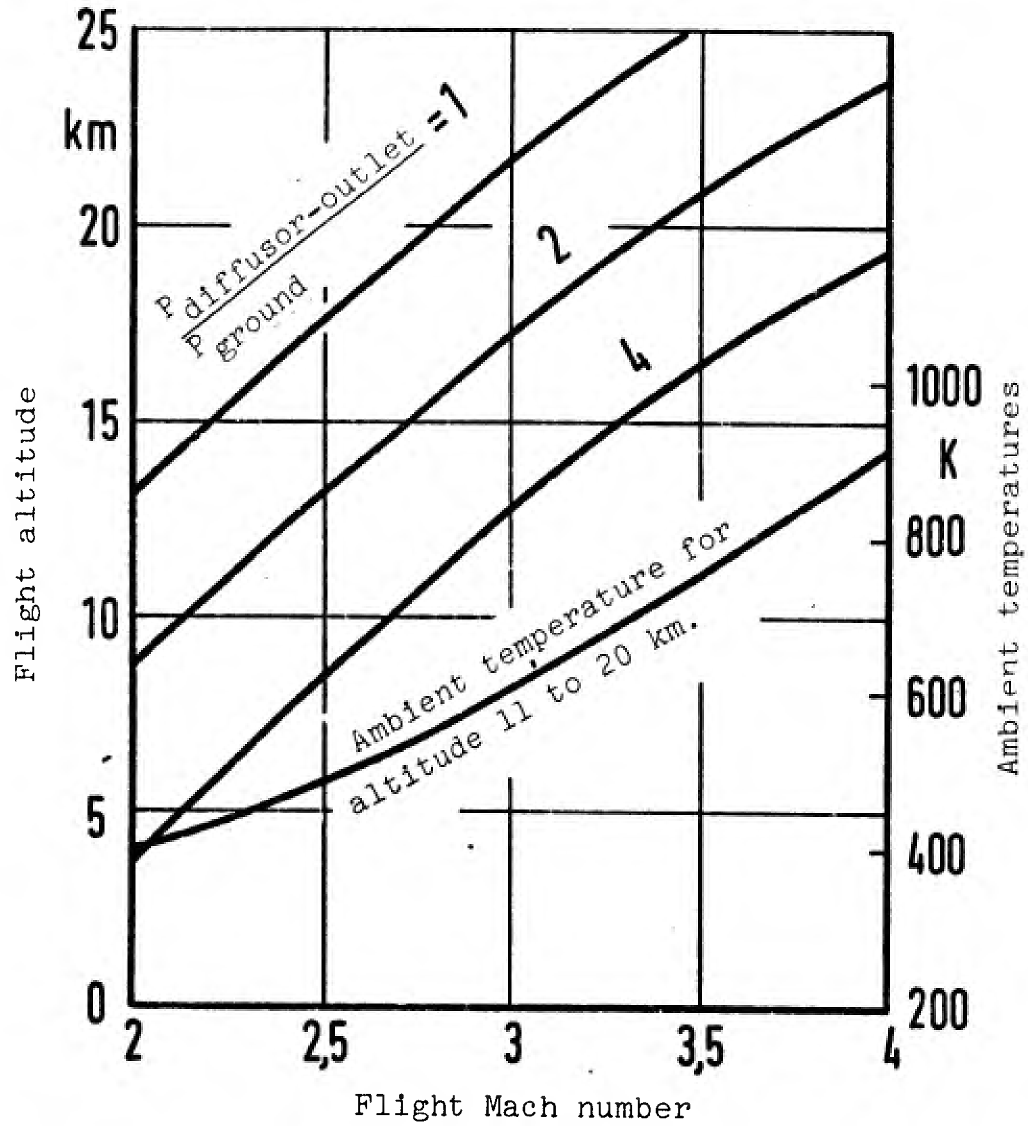

Figure 4. Flight altitude for constant diffusor exit pressure.

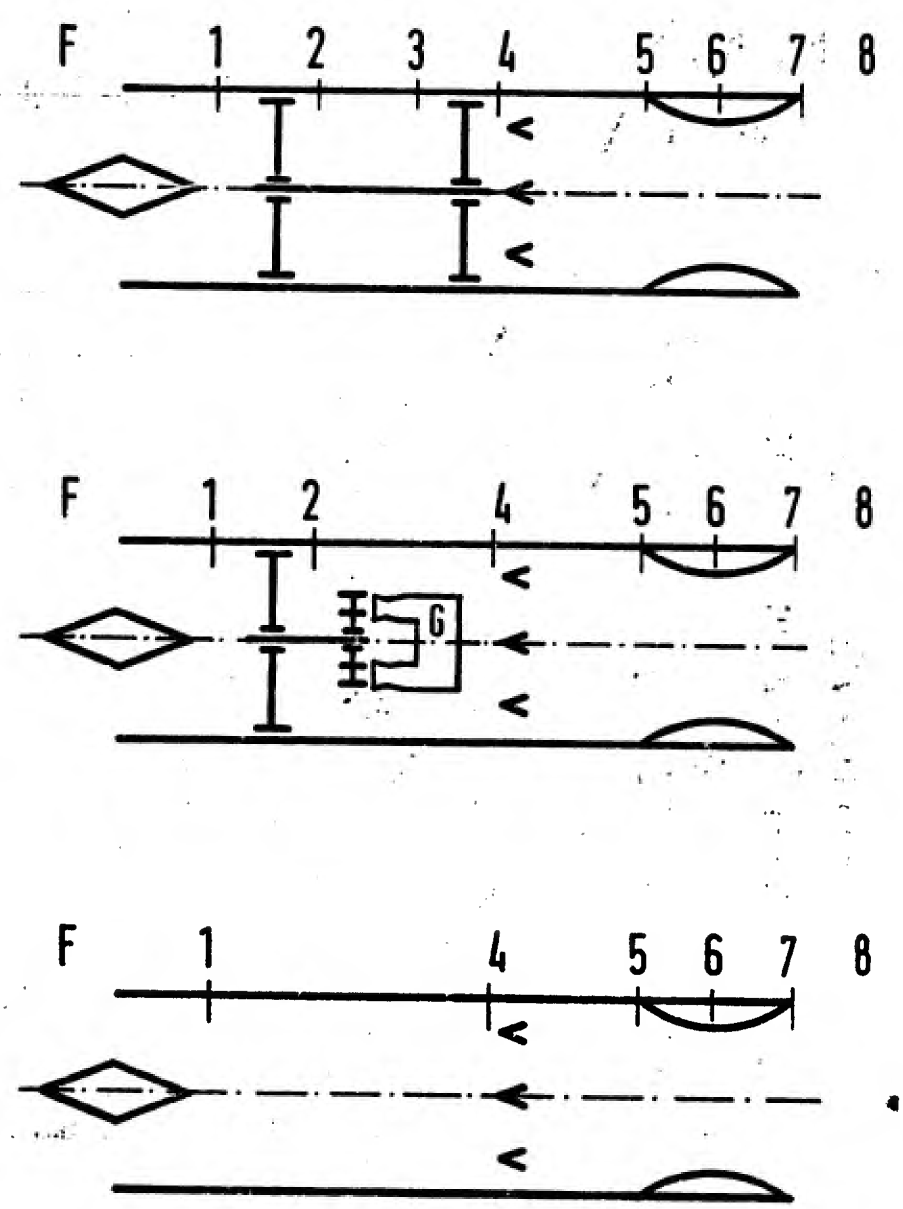

Figure 5. Engine types, schematic.

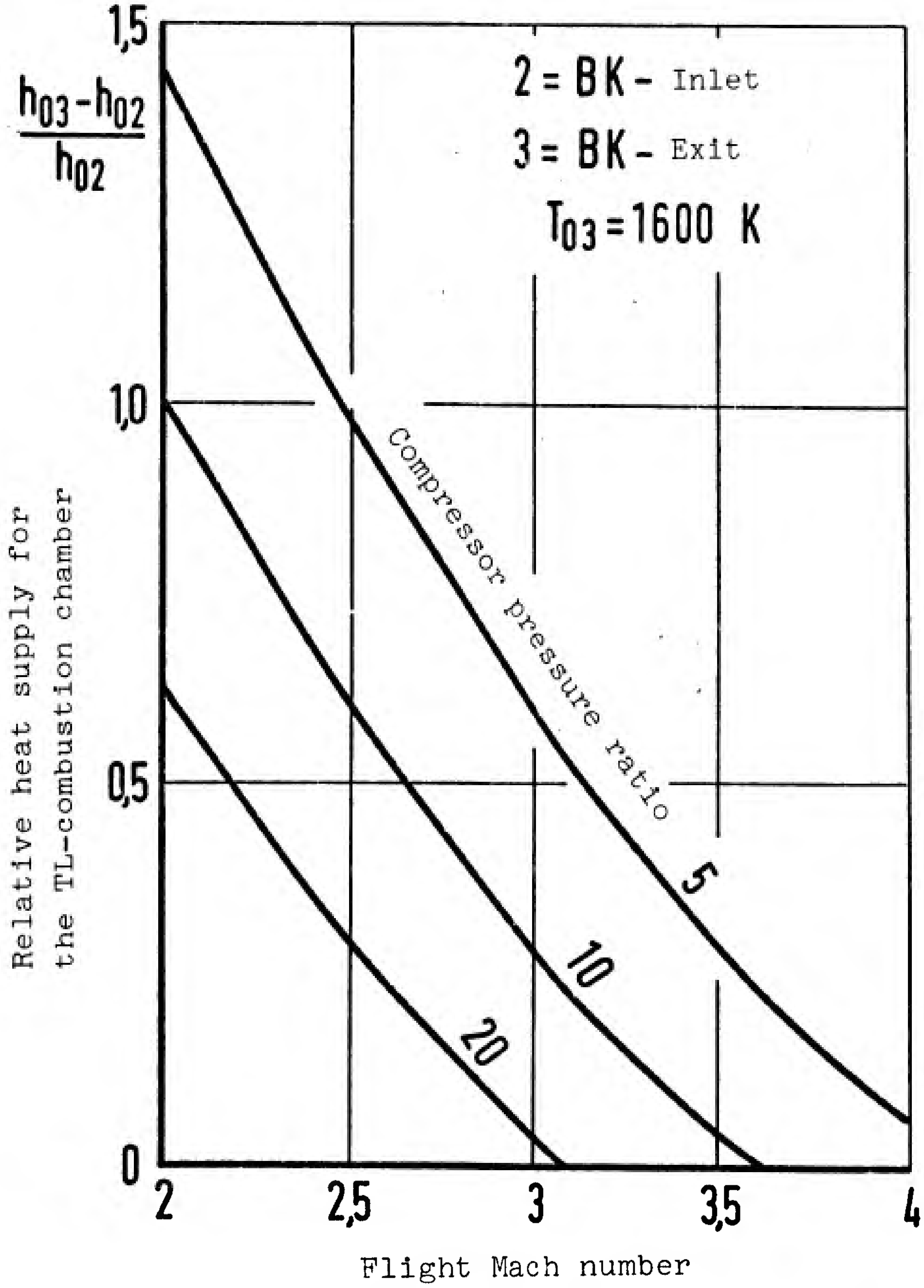

Figure 6. Possible heat supplied to the combustion chamber at 1600 K exit temperature.

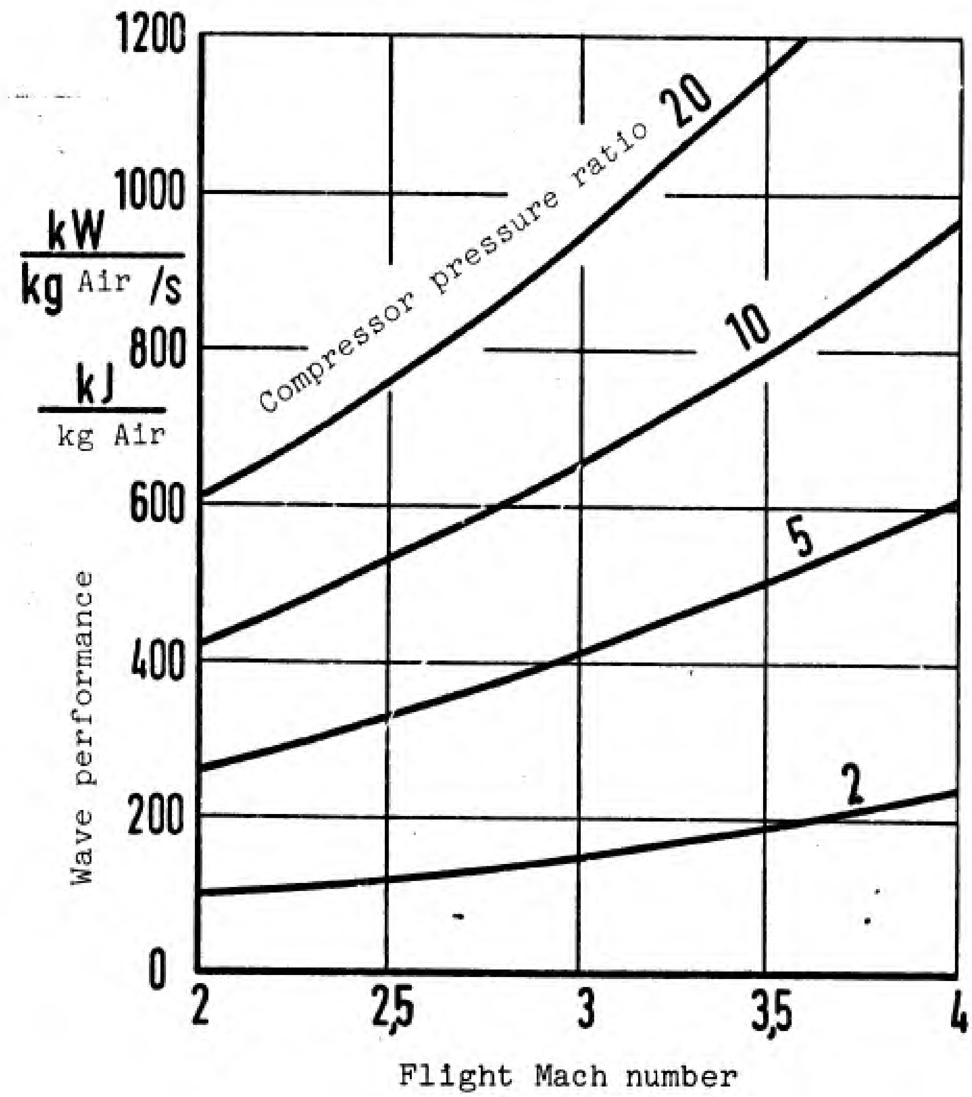

Figure 7. Required wave performance.

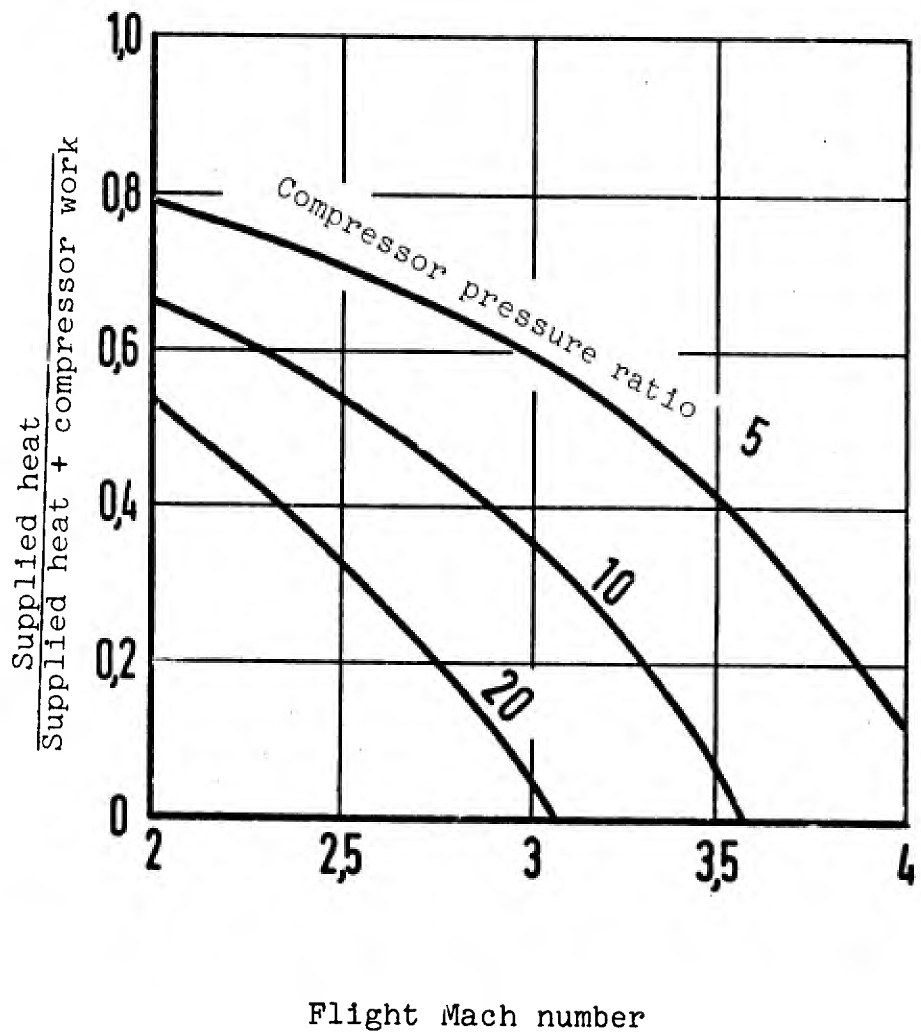

Figure 8. Fraction of heat supplied to the combustion chamber of the energy throughput through the combustion chamber.

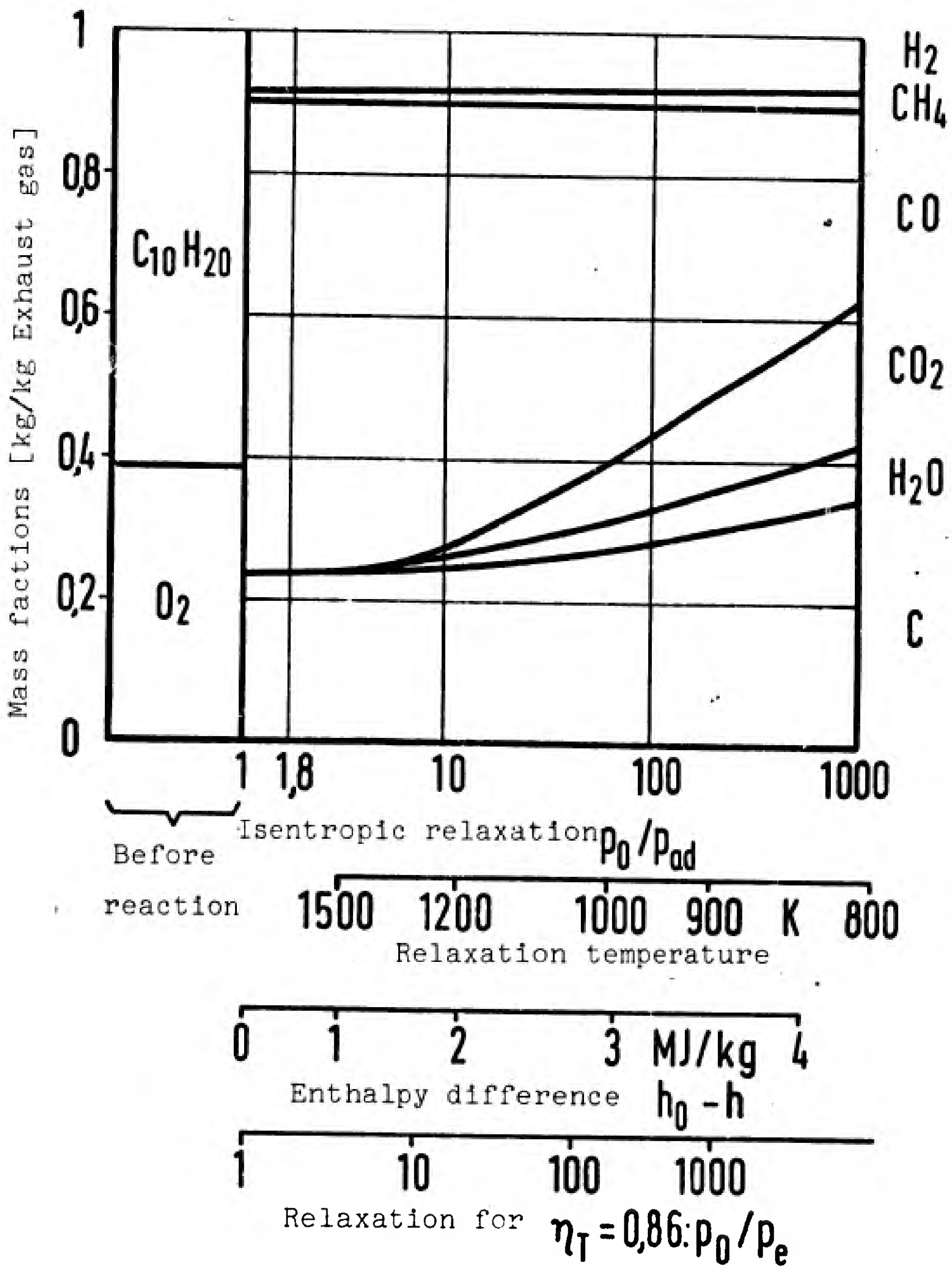

Figure 9. Gas generator, relaxation of the exhaust gas.

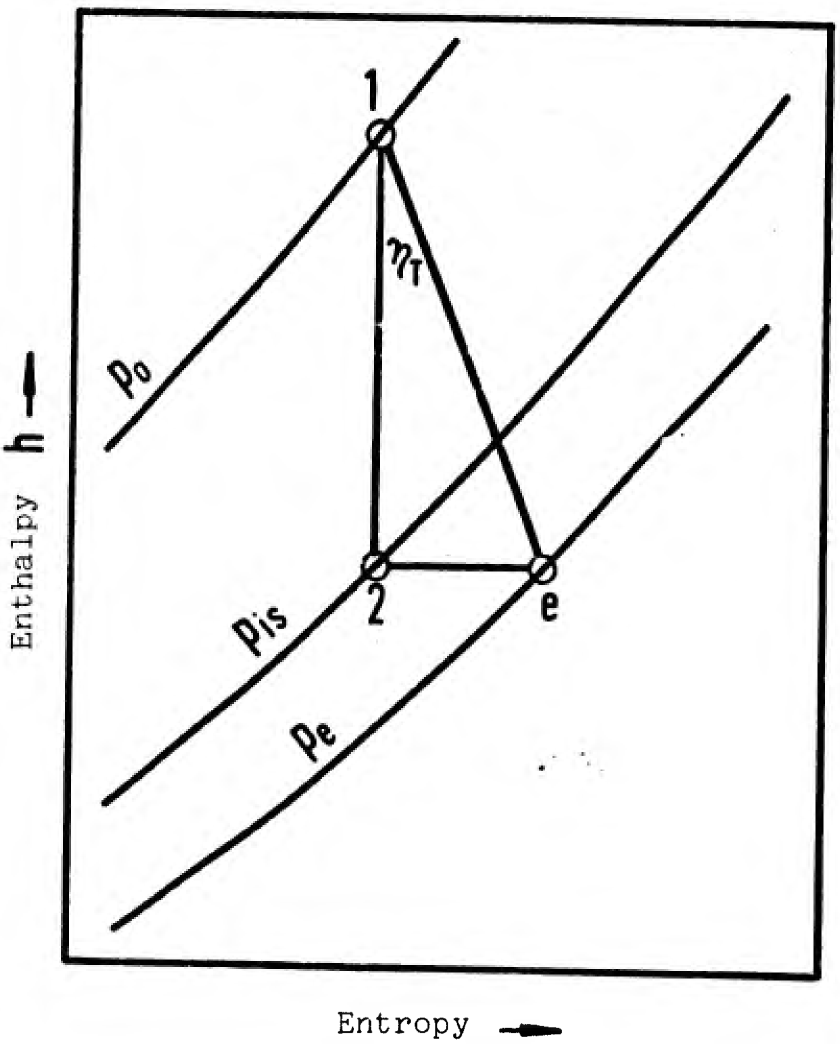

Figure 10. Determination of turbine pressure ratio.

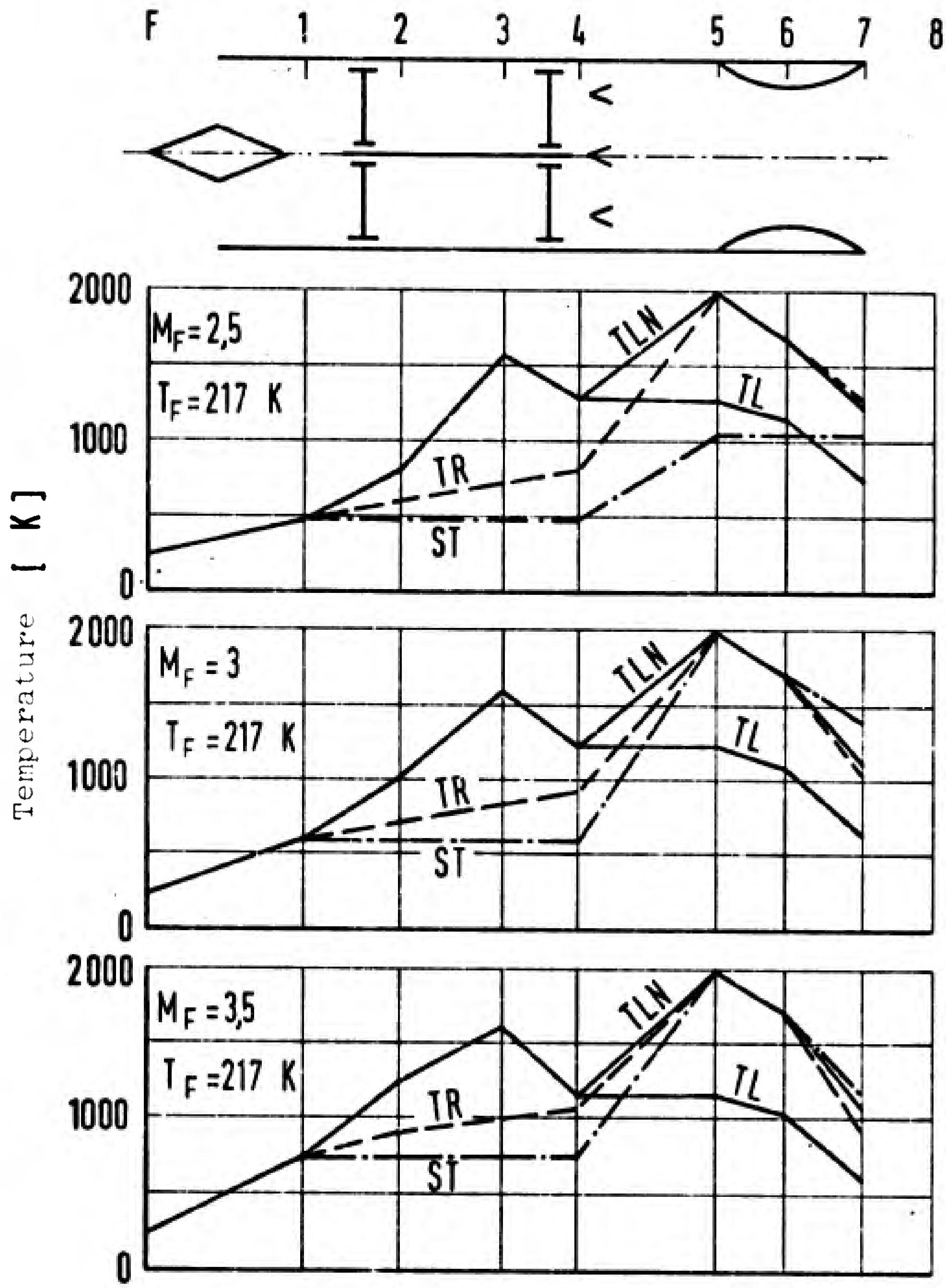

Figure 11. Temperature variations in the engines.

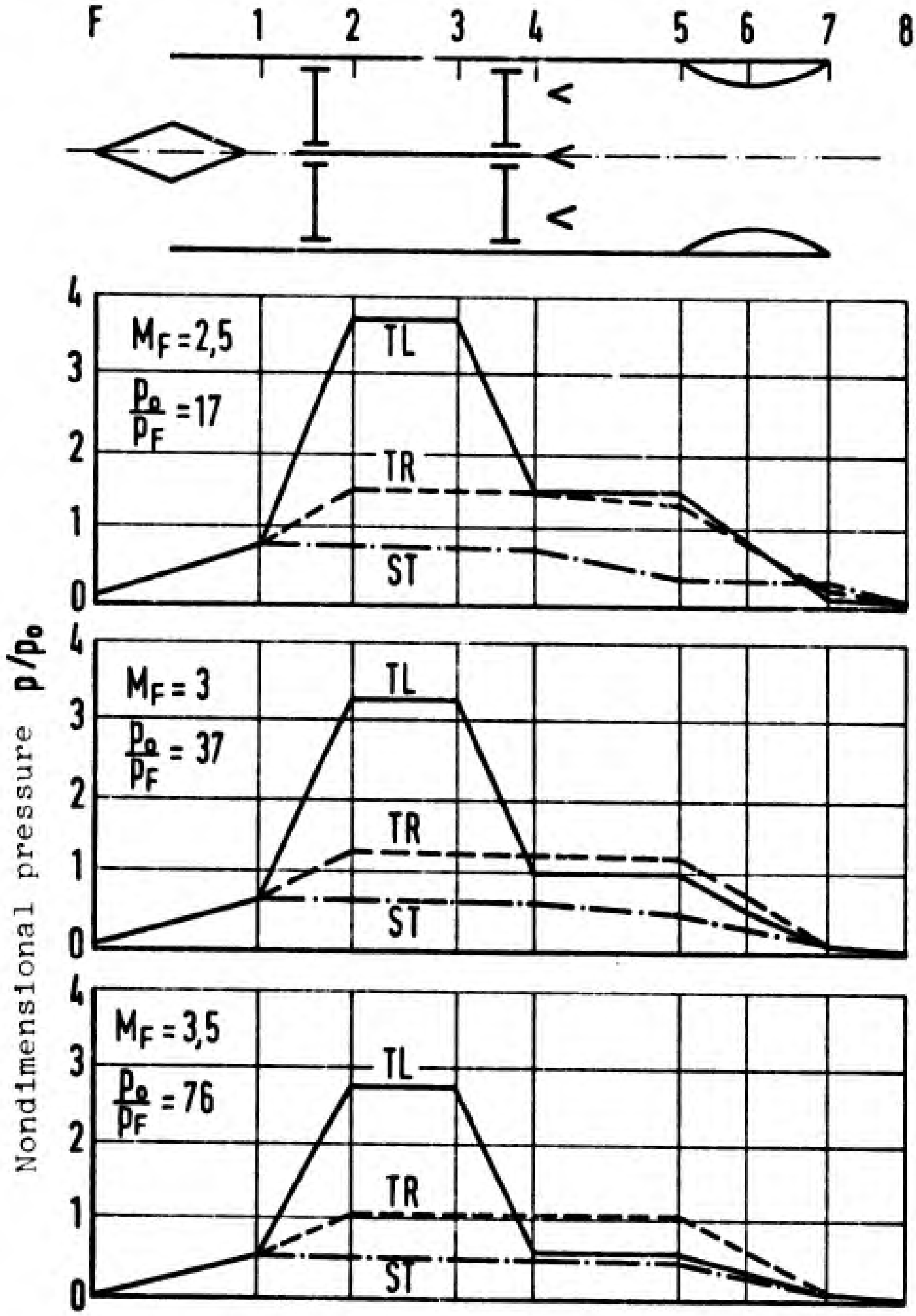

Figure 12. Pressure variations in the engines.

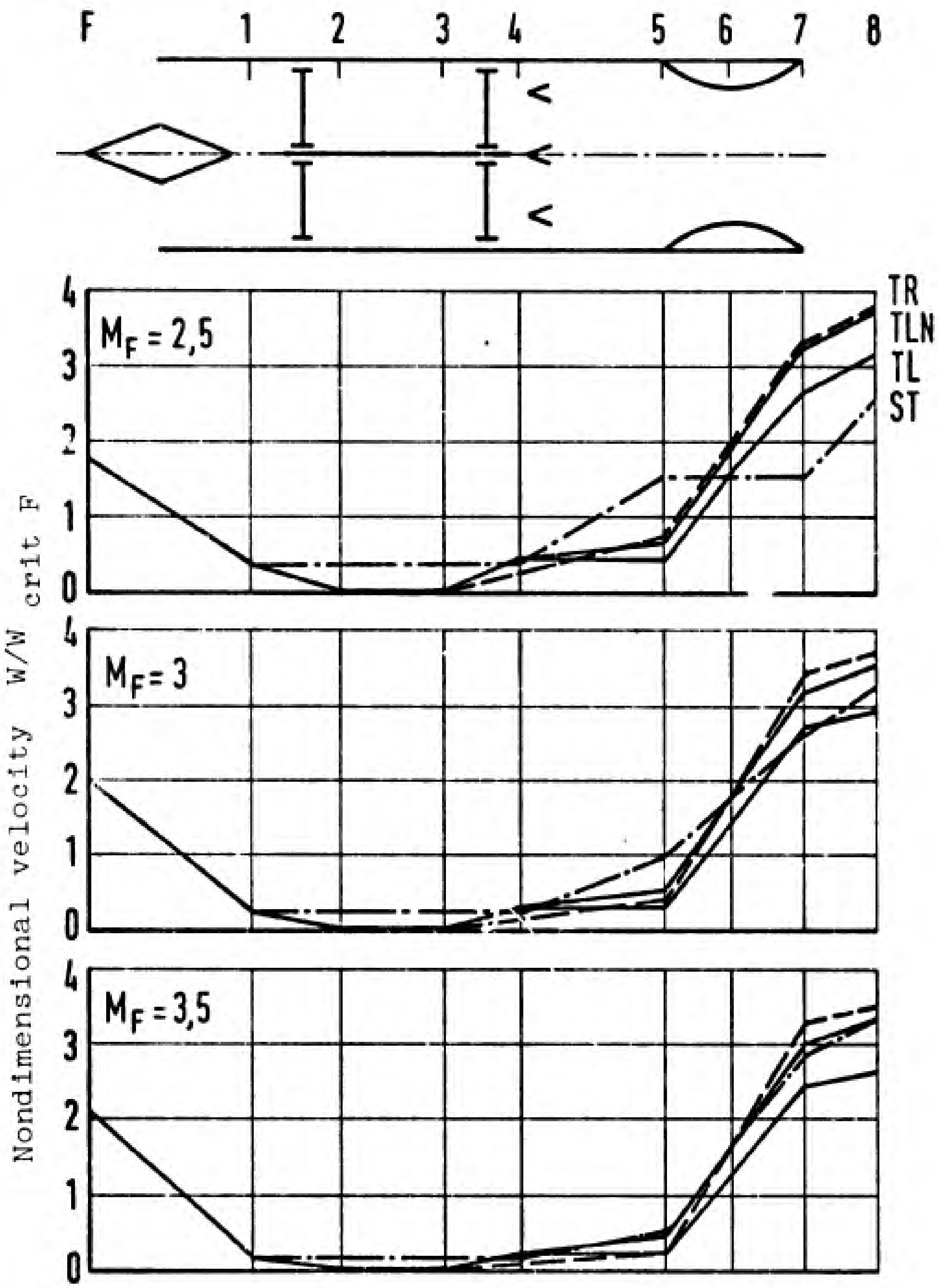

Figure 13. Velocity variations in the engines.

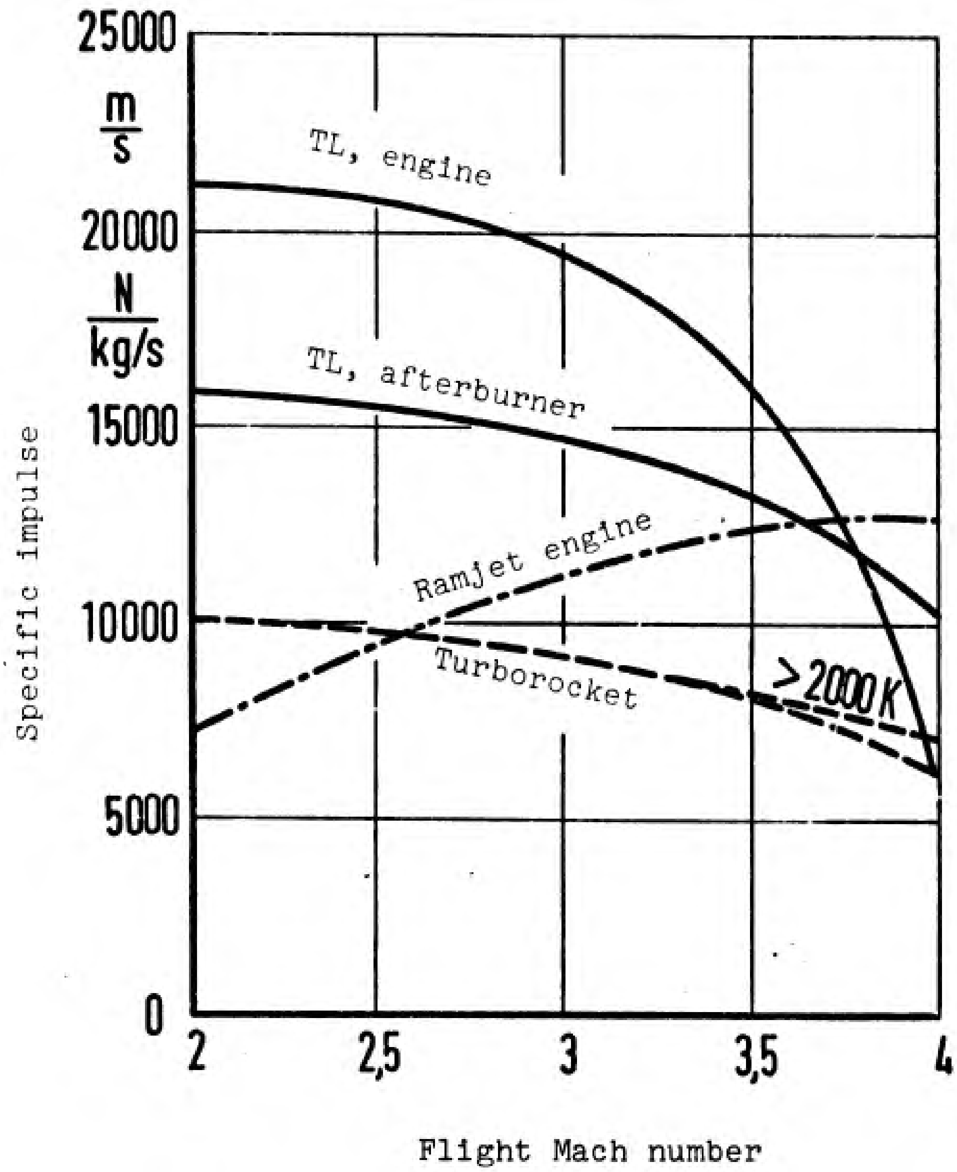

Figure 14. Specific impulse as a function of Mach number.

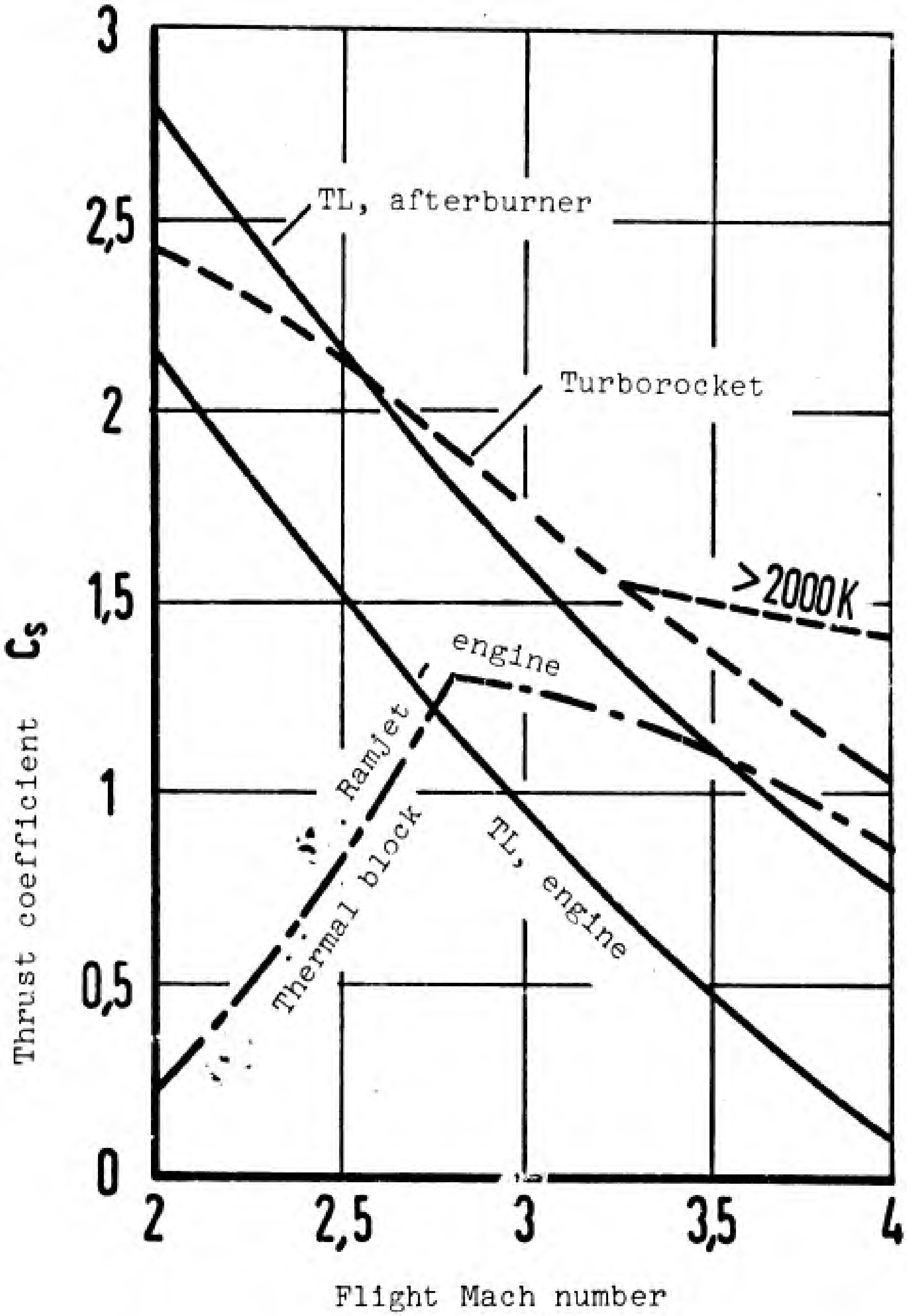

Figure 15. Thrust coefficient as a function of Mach number.

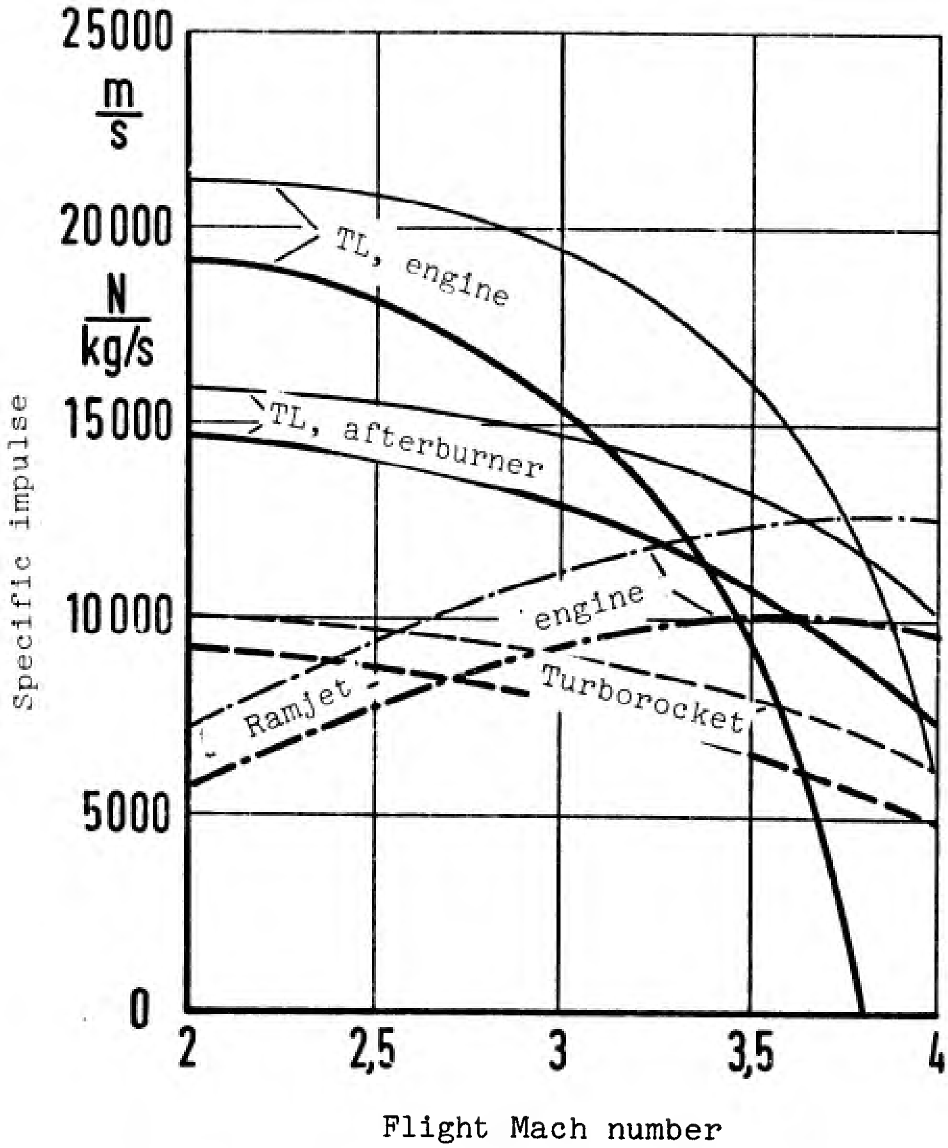

Figure 16. Comparison of net specific impulse.

Solid curve: $c_W = 0.2$ Thin lines: $c_W = 0$